

EP Performance Requirements for In-Situ Propellant Usage



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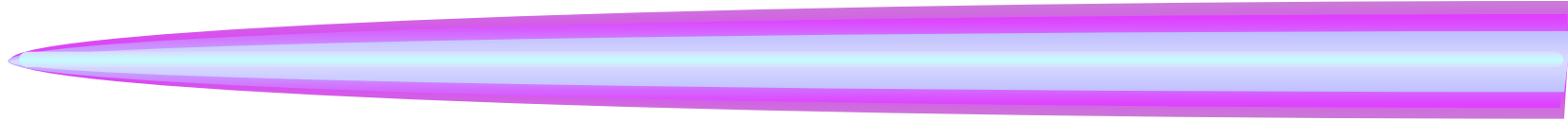
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Study Motivation



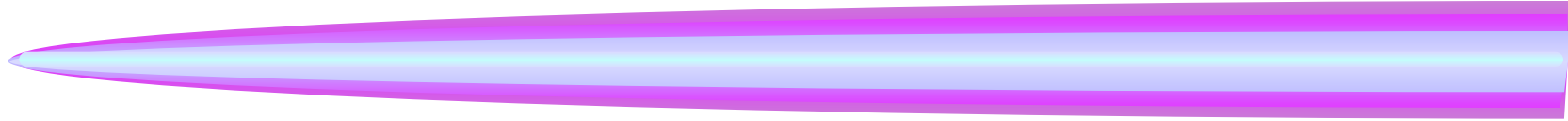
- For a “robust” exploration program, i.e.,
 - Continuous presence
 - Affordable operation
- In Situ propellant utilization must be considered.
 - In Situ is usually applied to low I_{sp} technologies
- The benefit, if any of In Situ utilization for electric propulsion (EP) systems should also be examined.
 - **Technology impacts**
 - Benefits

Analysis Approach



- Due to unknowns in electrodeless thruster performance, use “parametric thruster”
- Missions considered:
 - Lunar cargo
 - Essentially an orbital transfer mission
 - Constant ΔV of 8 km/s
 - Preliminary considerations for interplanetary (Mars)
- Incorporate mission analysis with thruster performance
- Look at necessary thruster efficiency for feasible missions

Lunar Cargo Mission Scenario

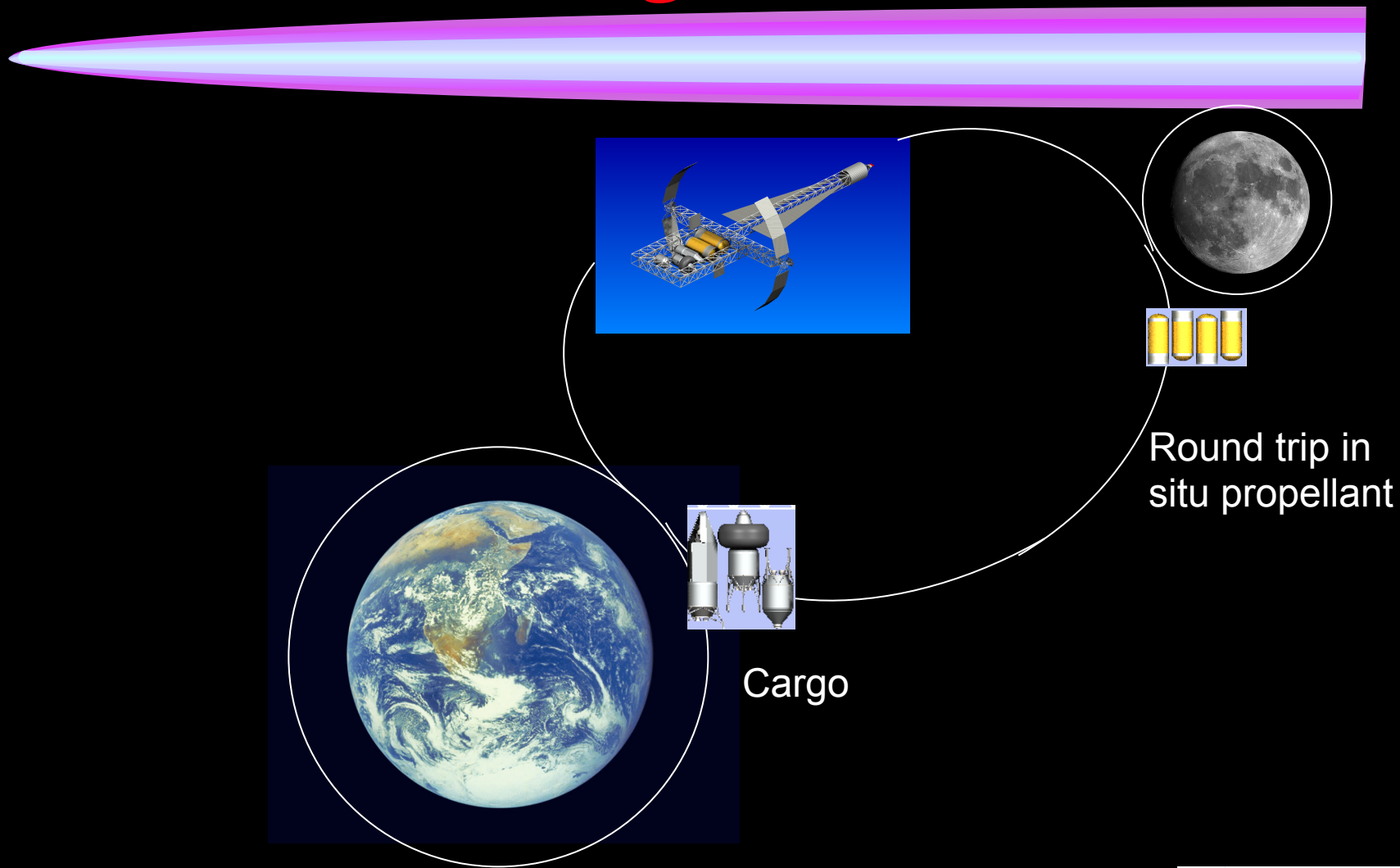


1. First Delivery:
2. Initial *reusable* cargo vehicle transports cargo to lunar orbit with Earth propellant
3. Cargo vehicle takes on return propellant and next outbound propellant load at Moon
4. Vehicle returns to receive next cargo load
5. Steady state operation proceeds using in situ propellant

Round trip time fixed at 300 d to allow for 1/year delivery rate

Mission No.	Payload Outbound	Propellant Source	Payload Inbound	Propellant Source
1	Lunar Cargo - M_l	Earth	Return propellant – M_{p2}	Moon
2, 3,	Lunar Cargo - M_l	Moon	Return propellant M_{p2}	Moon

Lunar Cargo Scenario

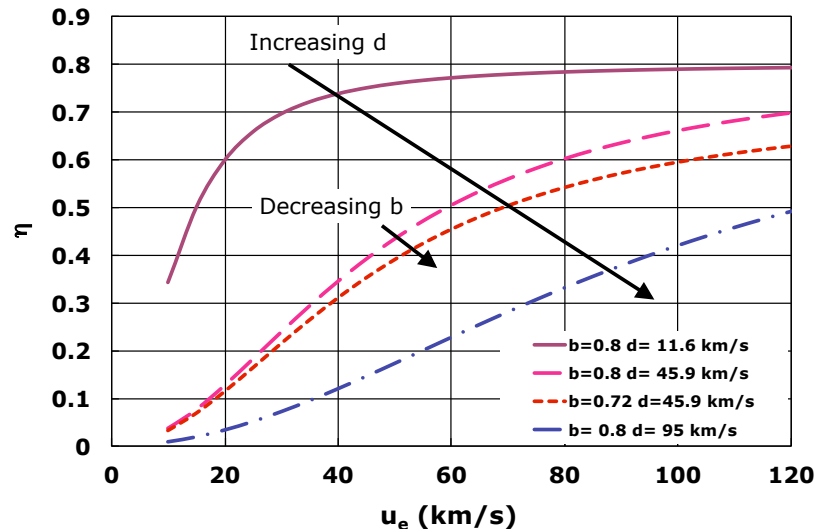


The “Parametric” Thruster

- η is a function of I_{sp} :

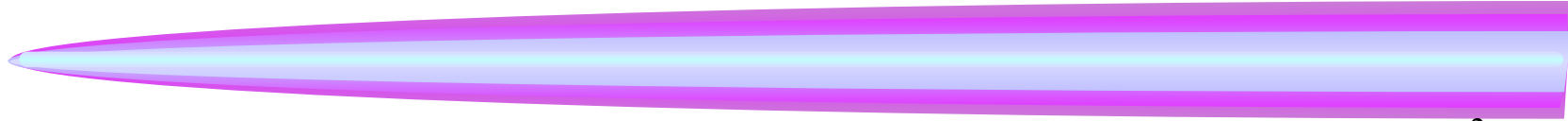
$$\eta = \frac{b I_{sp}^2}{d^2 + I_{sp}^2} \quad \text{or} \quad \frac{b u_e^2}{d^2 + u_e^2}$$

$$u_e = g_0 I_{sp}$$



- “b” term is essentially the efficiency of the acceleration process
 - Nozzle efficiency, for example
- “d” term is essentially the ionization cost of the plasma
 - Related to propellant type

Low Thrust Rocket Equation



Outbound (1):
$$e^{-\frac{\Delta V}{u_e}} = \frac{M_l}{M_i} = \frac{M_l + \alpha P_e}{M_l + \alpha P_e + M_p} \Rightarrow e^{-\frac{\Delta V}{u_{e1}}} = \frac{\mu + \frac{\alpha u_{e1}^2}{2\eta\tau_1}}{1 + \frac{\alpha u_{e1}^2}{2\eta\tau_1}}$$

Inbound (2):
$$e^{-\frac{\Delta V}{u_{e2}}} = \frac{M_{l2}}{M_{i2}} = \frac{M_{p1} + \alpha P_e}{M_{p1} + \alpha P_e + M_{p2}}$$

M_l = Payload Mass

M_p = Propellant Mass

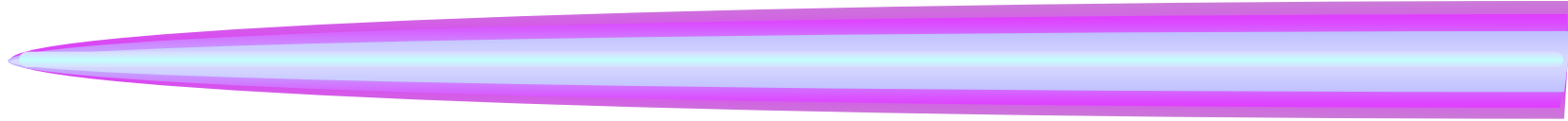
M_i = Initial Mass

α = Specific Mass (kg/kWe)

μ = Payload fraction = M_l/M_i

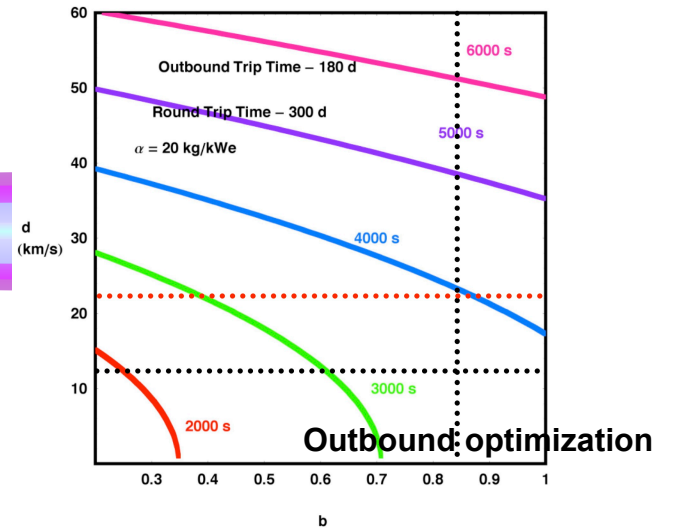
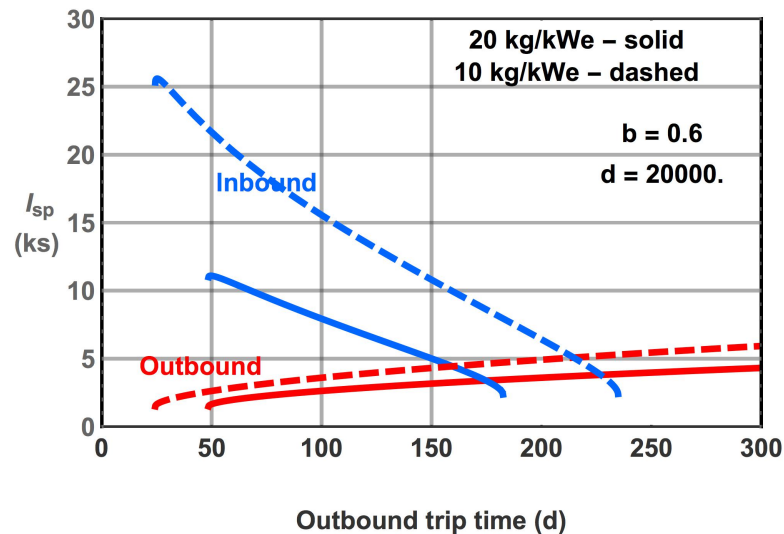
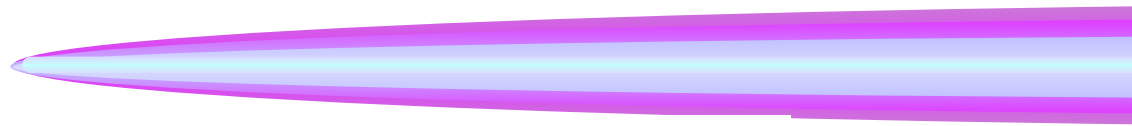
τ_1 = Outbound Thrusting time

Lunar Mission Calculations



- Optimize u_{e1} , power/payload for outbound mission
 - Functions of α , τ_1 , b , and d
- Calculate necessary return u_e based on outbound power level
- Use calculated power, return propellant to determine overall mission benefit
 - Results normalized to outbound payload mass
 - Express results in terms of efficiency (b , d)

Specific Impulse

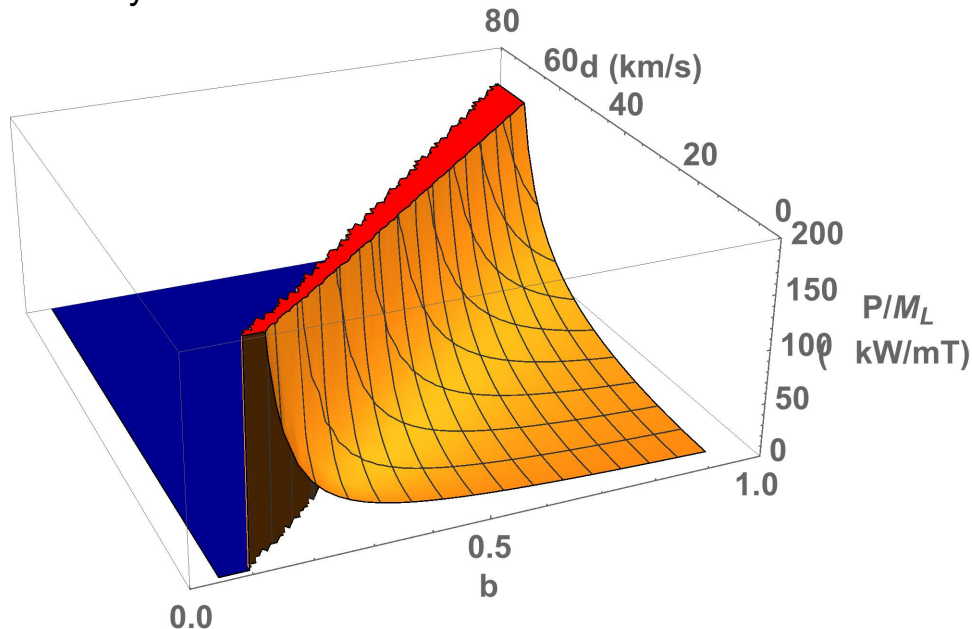


Propellant	I_{sp} (s)	b	d (km/s)
Xenon	1500-3300	0.86	11.9
Krypton	4000-7000	0.86	15.0
Argon	5000-8000	0.84	22.5
Xe Hall	1000-3000	0.87	14.1

- u_e outbound based on optimum for input trip time
 - Return $u_e = \text{outbound}$
 - Return power = outbound
- Power, propellant normalized to outbound payload mass

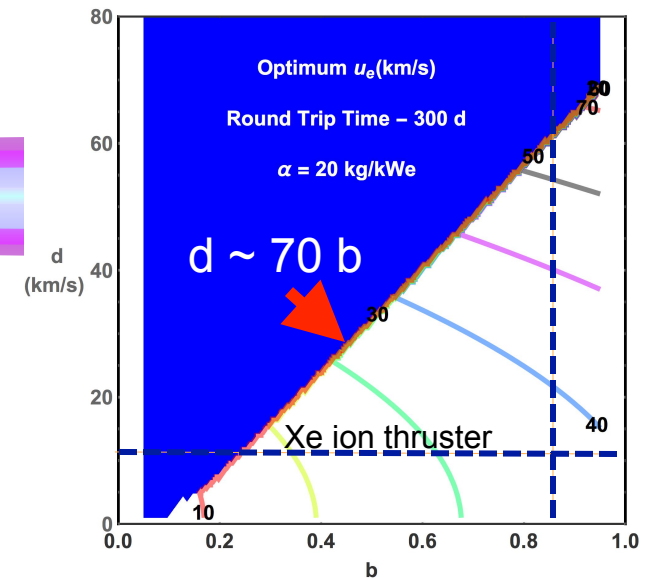
Return trip imposes more stringent limits on b , d

Power/Payload mass

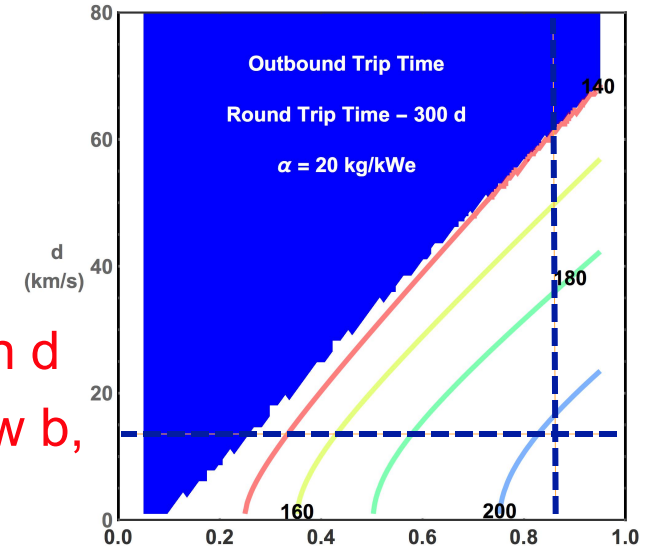


- Power: For 100 mt, 1 – 10 MWe at low b , high d
- Propellant requirements exceed payload at low b , high d

- Optimum u_e (km/s) contours

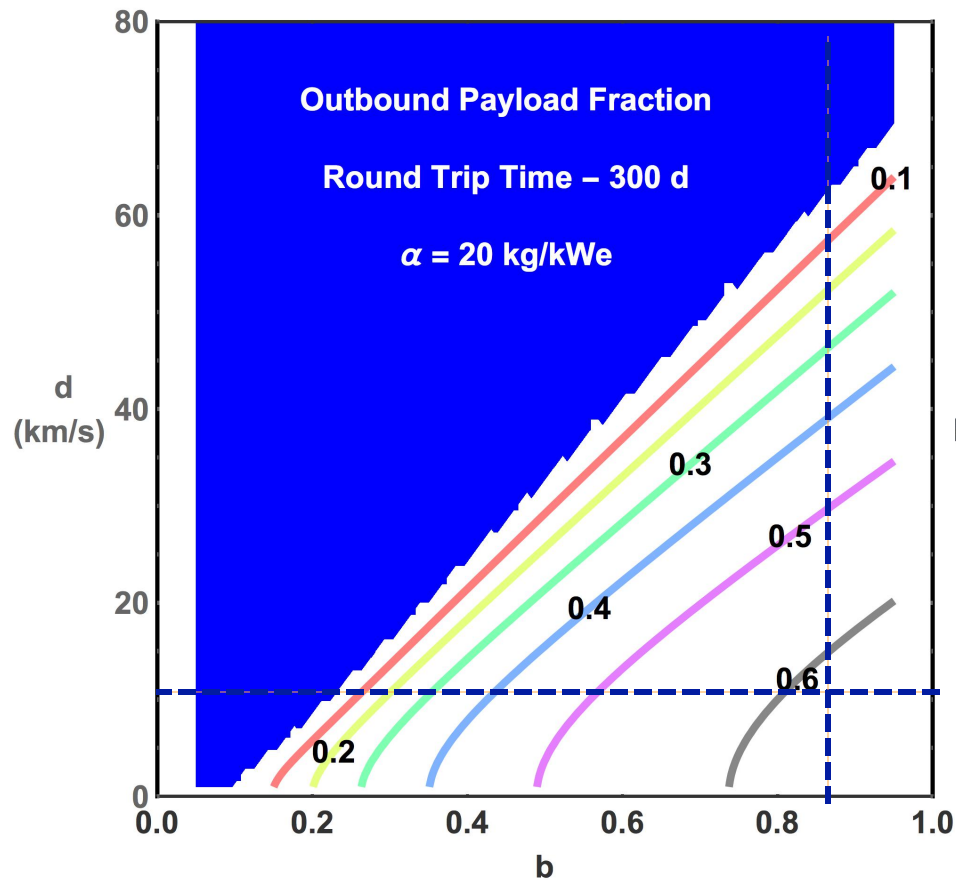
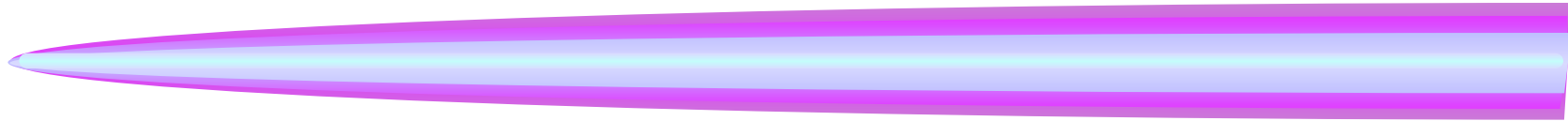


- Outbound trip time (d) contours

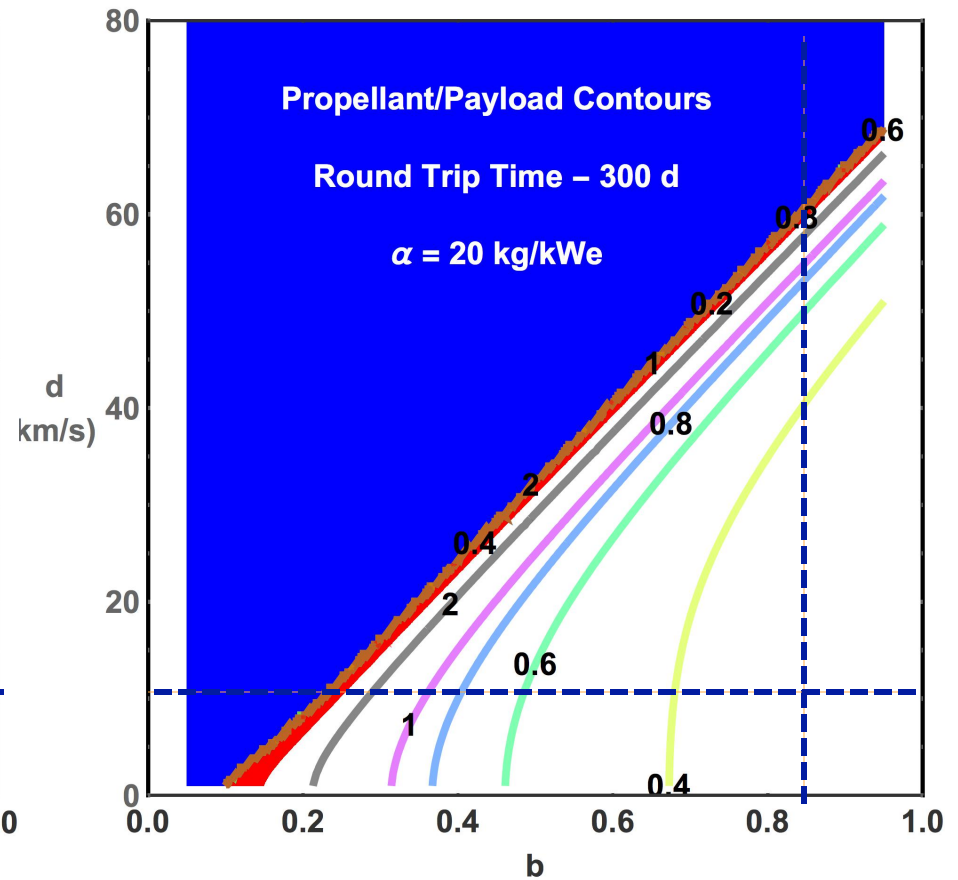


Mission performance

Low $b, d \rightarrow$ High Propellant

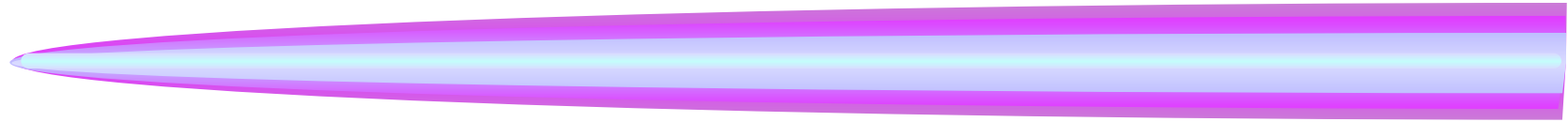


- Payload fraction



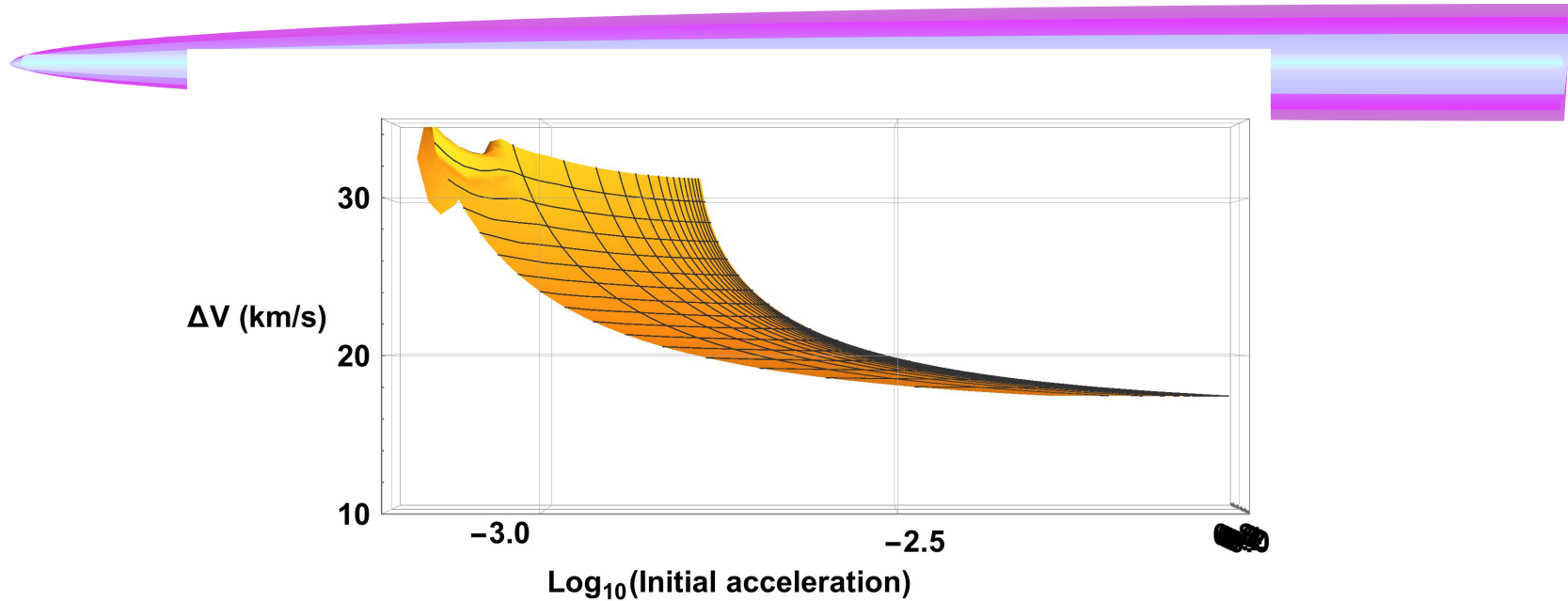
- Propellant/Payload ratio

Lunar Impacts



- $b < 0.5$, $d > 20$ km/s lead to strong increases in propellant, power requirements
 - Limiting space defined by $d/b \sim \text{constant}$
 - Constant will depend on system α , trip time requirements

Issues for interplanetary missions

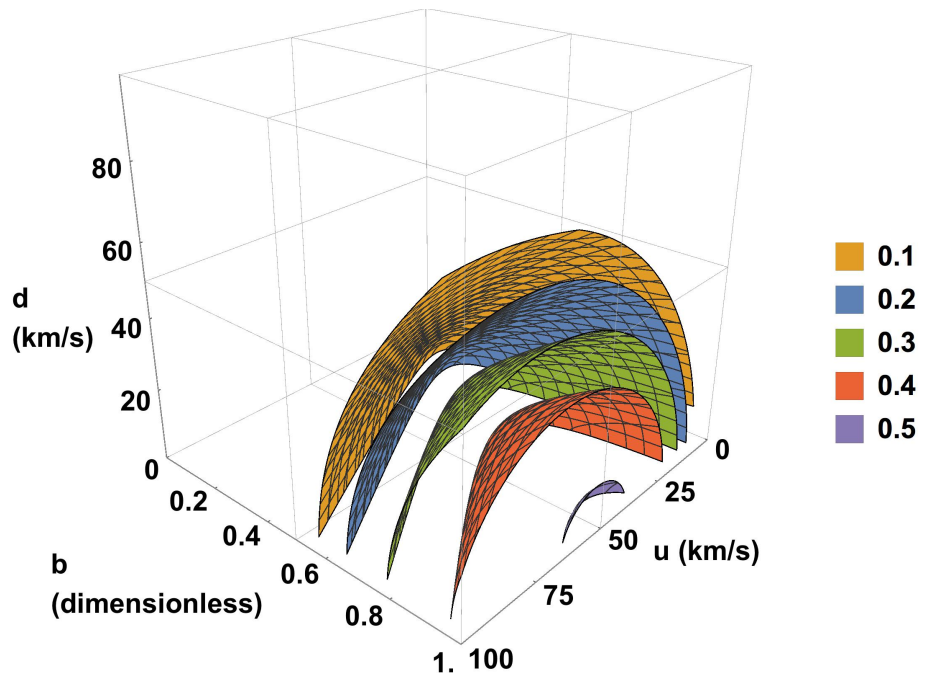


- Mission ΔV depends on system performance
- Complex optimization
- Refueling scenario is unclear
 - At planet?
 - At Earth?
 - Multiuse?

Preliminary Assessment

- One way mission sensitivity to thruster b , d
 - Payload fraction (μ_L) contours for range of u_e , b , d
 - 300 d interplanetary time, 10 kg/kWe propulsion system
- High (0.5) μ_L severely restricts u_e
 - $b < 0.5$, $d > 40$ km/s severely impacts performance

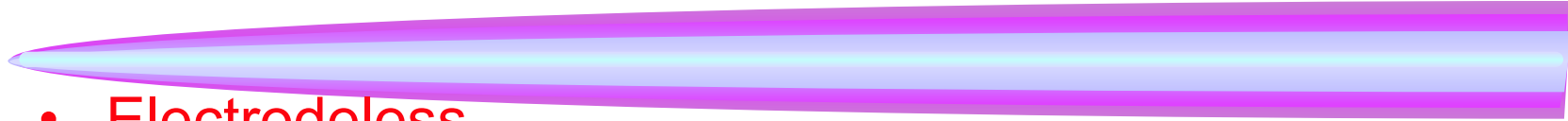
μ_L contours, 300 day Mars trip



Conclusions

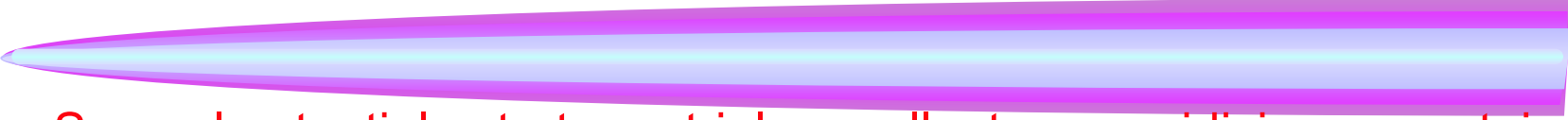
- The mission impact of varying thruster efficiency (as a result of using in situ propellants) has been examined parametrically for a lunar cargo and sample Mars mission
- Independent variables: efficiency parameters (b,d), trip time, specific mass, u_e
- Cases:
 - Lunar cargo (300 d round trip, 20 kg/kWe):
 - Constraint on $d/b \approx 70$
 - Strong power, propellant limits
 - Mars cargo (300 d one way, 10 kg/kWe):
 - $b < 0.5$, $d > 40$ km/s strongly increases power, propellant

Thruster Options



- **Electrodeless**
 - Performance is less well understood than “Conventional” thrusters
 - Pulsed Inductive Thruster (PIT)
 - No electrodes - inductive coil
 - Performance has been observed to be propellant dependent
 - Plasma wave concepts
 - Helicon, VASIMR,
 - Some (helicon) are based on electron heating – rf requirements independent of propellant
 - “Thermal” systems - propellant mass, excitation losses determine I_{sp} , η
 - FRC concepts

Electric Propulsion Issues with In Situ Propellants

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- Several potential extraterrestrial propellants are oxidizing or contain O_2
 - Lunar O_2 , H_2O
 - Martian CO_2
 - Molecular propellants degrade EP performance
 - Increased excitation losses lower efficiency (η)
 - Non optimal molecular weight affect I_{sp}
 - Multiple species can result in acceleration inefficiency
 - Thruster impacts
 - Corrosion of exposed metal anodes, cathodes, grids
 - Charge-to-mass effects on I_{sp}
 - Ionization and excitation losses decrease η
 - Changes in divergence with species can decrease η